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DESIGN AND TESTING OF A FLOX-METHANE THRUST CHAMBER ASSEMBLY

by Donald A. Peterson, Jerry M. Winter, Albert J. Pavli, and Arthur M. Shinn, Jr.

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16. Abstract

The results of a program to design and test fire a 600-pound-force (2670 N) FLOX-methane thrust-chamber assembly are presented. A characteristic exhaust velocity efficiency of 98.4 percent of theoretical equilibrium was obtained at the design oxidant-fuel ratio of 5.5 and a chamber pressure of 400 psia (2756 kN/m^2). Calculations indicate that peak delivered vacuum specific impulse of 395 seconds would be obtained at an oxidant-fuel ratio of 5.3. Conventional ablative materials were determined to have limited life in the throat region at chamber pressures above 200 psia (1378 kN/m^2). Successful operation at 400 psia (2756 kN/m^2) was obtained with an ablative chamber and a throat insert of ATJS graphite with no erosion or cracking after a 200-second continuous firing.

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SUMMARY

The results of a program to design and test a 600-pound-force (2670 N) passively cooled FLOX-methane thrust chamber assembly for high chamber pressure are presented. A 45-element oxidant-fuel-oxidant triplet injector was designed and test fired at 400 psia (2765 kN/m^2) chamber pressure. A characteristic exhaust velocity efficiency level of 98.4 percent of theoretical equilibrium was obtained at the design oxidant-fuel ratio of 5.5. Calculations using experimental η_{C*} and assuming a nozzle expansion area ratio of 60 indicate that the peak delivered vacuum specific impulse of 395 seconds would be obtained at an oxidant-fuel ratio of 5.3.

Conventional ablative materials were determined to have limited lifetimes when chamber pressures exceed 200 psia (1376 kN/m^2). Thirteen throat inserts of refractory metals, composites, and a transpirant-cooled concept were tested at 400 psia (2756 kN/m^2) chamber pressure. Successful operation at 400 psia (2756 kN/m^2) chamber pressure was obtained using a throat insert of ATJS graphite with no erosion or cracking after a 200-second continuous firing.

Operation at an oxidant-fuel ratio of 4.1 resulted in minimum throat area change due either to oxidation or carbon deposition which indicates that an optimized engine could be designed with the use of an oxidant-fuel-ratio zoned injector.

INTRODUCTION

Mission analysis for deep space applications has indicated a need for propulsion systems with high specific impulse together with propellant space storage capability. Liquid FLOX (82.6 percent fluorine and 17.4 percent oxygen) and liquid methane are leading candidates for these applications. This combination has the highest theoretical specific impulse of all the hydrocarbon fuels (418 sec). FLOX and methane are readily stored in space because of high boiling points, compared to hydrogen. This propellant combination also has a high bulk density due to the high percentage of fluorine in the oxidizer and operation at an oxidant-fuel ratio (O/F) of 5.50. An O/F of 5.50 provides peak theoretical kinetic specific impulse at an expansion area ratio of 40 (ref. 1).

However, two facts must be considered before these advantages are realized in an actual engine system. First, it is essential that both of the propellants be thoroughly mixed and completely reacted within the combustion chamber to obtain the theoretical maximum impulse. Secondly, of all the liquid petroleum gas fuels, methane has the lowest heat capacity as a liquid. Only at cooling jacket pressures above critical (673 psia (4640 kN/m^2)), where problems of boiling heat transfer can be avoided, would methane be attractive as regenerative coolant (ref. 2). Initial studies indicated a regenerative engine would be feasible at 400 psia (2756 kN/m^2) chamber pressure. Therefore, the need existed to evaluate the performance capability of these propellants and to determine if passively cooled chambers are a feasible alternative to regenerative engines at chamber pressures up to 400 psia (2756 kN/m^2). Ablative chambers and nozzles have been successful at 100 psia (689 kN/m^2) chamber pressure for a FLOX-propane engine (refs. 3 and 4) while throat inserts were successfully used at 200 psia (1378 kN/m^2) pressure (ref. 3).

A program was initiated to develop an ablative FLOX-methane thrust chamber assembly for operation at chamber pressures of 200 psia (1378 kN/m^2) and 400 psia (2756 kN/m^2). The oxidant-fuel ratio to deliver peak impulse at an expansion area ratio of 60 was also to be determined. A characteristic exhaust velocity efficiency of over 95 percent of theoretical shifting equilibrium was set as a goal to be demonstrated with stable combustion. A continuous run duration of 200 seconds was also sought as a demonstration of passive cooling feasibility. Ablative materials and throat inserts were tested to determine the suitability of these materials as an alternative cooling approach to the more complex and possibly marginal technique of regenerative cooling.

Another objective of the program was to determine what oxidant-fuel ratio would result in an acceptably small throat area change due either to oxidation or carbon deposition. If this experimentally determined O/F resulted in a small impulse penalty, a performance optimized engine might then be designed.

The nominal operating conditions for the thrust chamber included an initial throat diameter of 1.2 inches (3.05 cm), an O/F range of 3.5 to 7.0, chamber pressures of 200 and 400 psia (1378 and 2756 kN/m^2), and an expansion area ratio of approximately 2.0 (sea-level operation). The thrust generated was 300 to 600 pounds force (1335 to 2670 N) when corrected to vacuum conditions. The predicted peak impulse was determined by using the experimental heat release efficiency with the theoretical kinetic data at an expansion area ratio of 60 to 1.

APPARATUS

THRUST CHAMBER DESIGN

The nominal design conditions were a throat diameter of 1.2 inches (3.05 cm), a characteristic chamber length L* of 33 inches (84 cm), and a chamber contraction ratio of 3.0. The contraction ratio of 3.0 was selected to prevent excessive chamber erosion based on the results of reference 3. The design O/F of 5.5 was selected as the value which provided maximum theoretical vacuum kinetic specific impulse for an expansion area ratio 40 nozzle (ref. 1) Chamber pressures of 200 and 400 psia (1378 and 2756 kN/m^2) were selected to explore the range above that used for pressure fed engines (100 psia (689 kN/m^2)).

Water-cooled chambers with both water-cooled and heat sink nozzles were used to obtain injector performance measurements.

Ablative material selection was based upon results obtained at 100 psia chamber pressure with FLOX-propane (ref. 3). Various representative types of throat insert materials were selected to establish relative merit in the FLOX-methane combustion environment.

Injectors

The oxidant-fuel ratio required for theoretical maximum specific impulse with FLOX-methane is near stoichiometric if HF and CO are assumed as the reaction products. Both propellants must be thoroughly mixed, therefore, to assure utilization of each molecule in the combustion-reaction process.

The general analysis and design methods used for FLOX-propane (ref. 3) were also applied to FLOX-methane. The vaporization criteria of Priem in addition to the following design techniques were found useful with FLOX-propane. Oxidant-fuel-oxidant triplets were used to keep the fuel and oxidant hole sizes similar. Mutually perpendicular elements were used to provide good propellant mixing between the fans. Small diameter (0.0135 to 0.021 in. (0.0343 to 0.0534 cm)) holes with attendant small stream diameters were used to assure complete vaporization in the L* 33-inch (84 cm) engine. All injectors were designed for liquid-liquid injection as would be appropriate in nonregenerative cooled engines. The design velocity ratio (oxidant/fuel) of 0.58 was also similar to that used for FLOX-propane. Table I lists the design values for each of the injectors. No attempt was made to maintain low injector pressure drops since the hypothetical application was a pump-fed engine. All the injectors were constructed of 6061-T6 aluminum material. Face cooling was accomplished by cross-drilling the passages for the oxidant supply.

TABLE I. - INJECTOR DESIGN VALUES

[Injector element discharge coefficient assumed 0.7 for pressure-drop calculation; oxidant weight flow, 0.924 lb/sec (0.419 kg/sec) and fuel weight flow, 0.168 lb/sec (0.0762 kg/sec) for chamber pressure of 200 psia (1378 kN/m^2); oxidant weight flow, 1.848 lb/sec (0.838 kg/sec) and fuel weight flow, 0.336 lb/sec (0.152 kg/sec) for chamber pressure of 400 psia (2756 kN/m^2); ratio of oxidant injection velocity to fuel injection velocity, V_{ox}/V_f, 0.58.]

Injector	Chamber pressure level, P_c		Oxidant							Fuel						
	psia	(kN/m^2)abs	Number of holes	Hole diameter		Pressure drop, ΔP_{ox}		Injection velocity, V_{ox}		Number of holes	Hole diameter		Pressure drop, ΔP_f		Injection velocity, V_f	
				in.	cm	psi	kN/m^2	ft/sec	m/sec		in.	cm	psi	kN/m^2	ft/sec	m/sec
1	200	1378	74	0.0156	0.0396	201	1385	101	30.8	37	0.0135	0.0343	178	1226	173	52.7
2	400	2756	74	.021	.053	256	1764	112	34.1	37	.0180	.0457	226	1557	193	58.8
3	400	2756	90	.021	.053	173	1192	92	28.0	45	.0180	.0457	153	1054	159	48.5

Injector	Diameter of -			
	Oxidant holes		Fuel holes	
	in.	cm	in.	cm
1	0.0156	0.0396	0.0135	0.0343
2	.021	.053	.018	.046

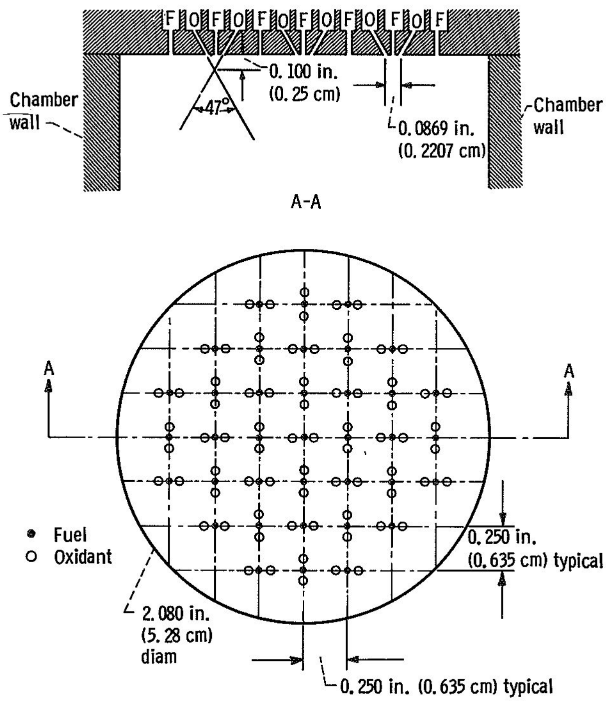


Figure 1. - Injectors 1 (for chamber pressure of 200 psia; 1378 kN/m^2) and 2 (for chamber pressure of 400 psia; 2756 kN/m^2). Oxidant-on-fuel square pattern triplet; 74 oxidant and 37 fuel holes for each injector.

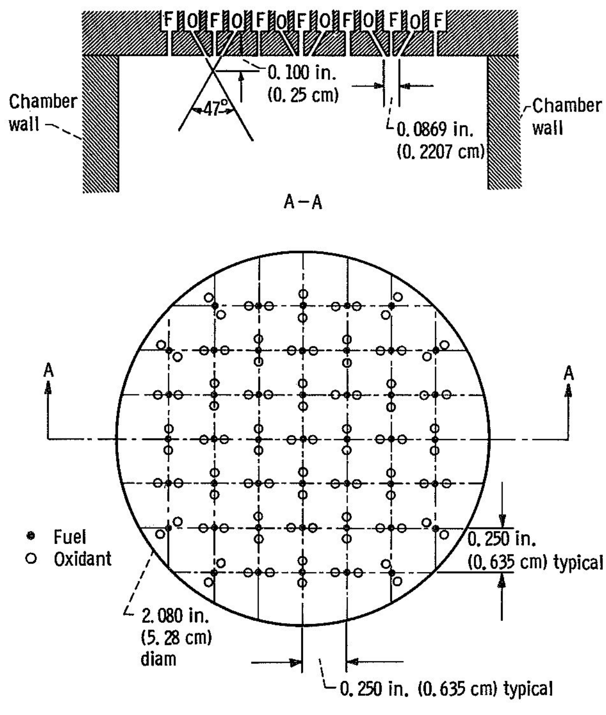


Figure 2. - Injector 3 (for chamber pressure of 400 psia; 2756 kN/m^2). Modified oxidant-on-fuel square pattern triplet; 90 oxidant holes 0.021 inch (0.053 cm) in diameter; 45 fuel holes 0.018 inch (0.046 cm) in diameter.

Injectors 1 and 2 were designed for use at 200 and 400 psia (1378 and 2756 kN/m^2) chamber pressure, respectively. The pattern, shown in figure 1, was the same for both injectors. The design O/F was 5.5.

Injector 3 used the same basic pattern with eight additional elements as indicated in figure 2. Element placement was selected to fill voids in the pattern thus preventing recirculation and counterflow. The added elements were radially oriented to form impingement fans parallel to the wall for better ablative compatibility.

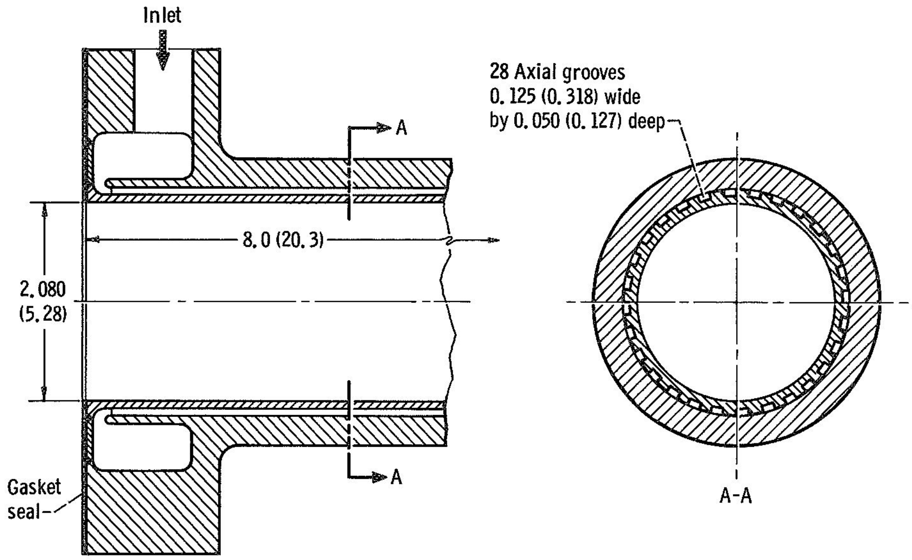
Chambers and Nozzles

The water-cooled chamber design for operation at 200 psia (1378 kN/m^2) chamber pressure is shown on figure 3(a). A water flow rate of 4.31 pounds per second (1.95 kg/sec) resulted in an overall pressure drop of 140 psi (965 kN/m^2) with a velocity of 56.7 feet per second (17.2 m/sec) in the coolant passages. Figure 3(b) shows the water-cooled chamber nozzle used at 200 psia (1378 kN/m^2) chamber pressure. A water flow rate of 3.81 pounds per second (1.73 kg/sec) resulted in an overall pressure drop of 150 psi (1030 kN/m^2) with a maximum velocity of 36.3 feet per second (11 m/sec) in the coolant passages. These designs were intended to operate satisfactorily at a heat flux of 6 Btu per square inch per second (9800 kW/m^2). The thrust chamber assembly described in figure 3 had an L^* of 33 inches.

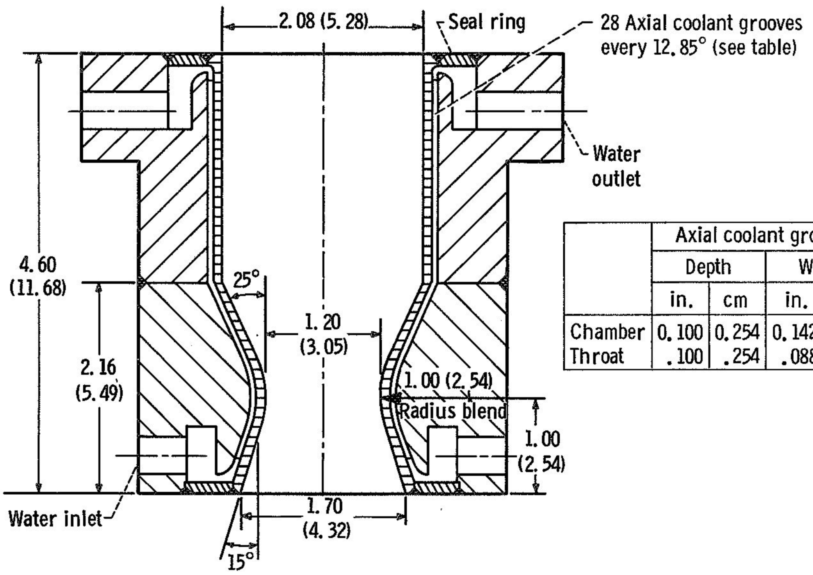
Figure 4(a) is a drawing of the copper water-cooled chamber, designed for operation at 400 psia (2756 kN/m^2) chamber pressure. A water flow rate of 6 pounds per second (2.72 kg/sec) resulted in an overall pressure drop of 80 psi (551 kN/m^2) with a velocity of 21 feet per second (6.4 m/sec) in the coolant passages. The chamber was designed to operate satisfactorily at a heat flux of 10 Btu per square inch per second (16 300 kW/m^2).

Figure 4(b) is a drawing of the copper heat-sink chamber nozzle for use at 400 psia (2756 kN/m^2) chamber pressure. The copper design was intended for only 2 to 4 seconds of engine operation, which time was adequate for determining engine performance. The assembly, combining chamber and nozzle of figure 4, had an L^* of 31 inches.

The ablative chamber-nozzle design of figure 5(a) was the same as that used for FLOX-propane at 100 psia (689 kN/m^2) chamber pressure (hardware was on hand after completion of work reported in ref. 3). The General Electric REKAP theoretical computer method of reference 5 was used for the original design. Thicknesses were based on a 300-second continuous firing at 100 psia (689 kN/m^2). The material was graphite cloth-phenolic (FM5064, lay up 60° to nozzle centerline), the best ablative material tested in reference 3.



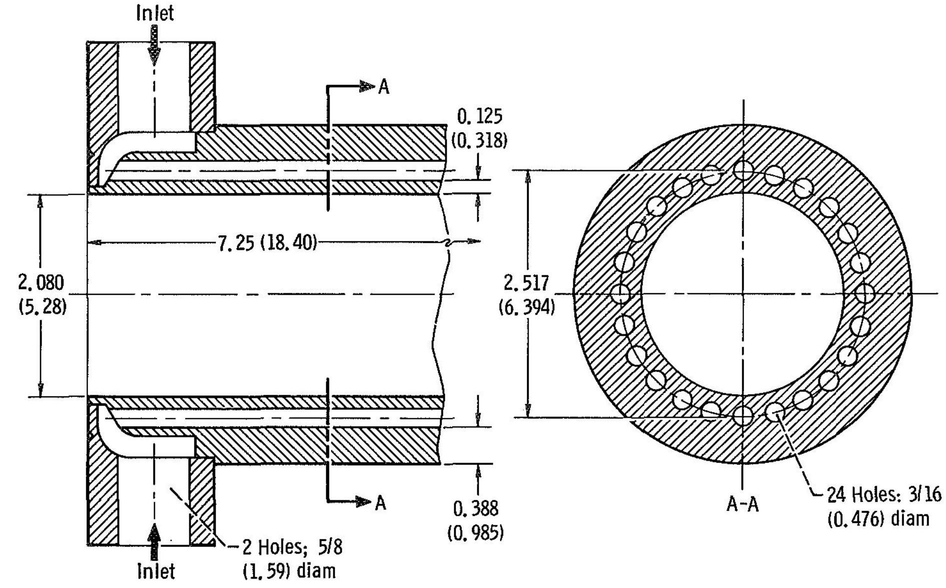
(a) Chamber.



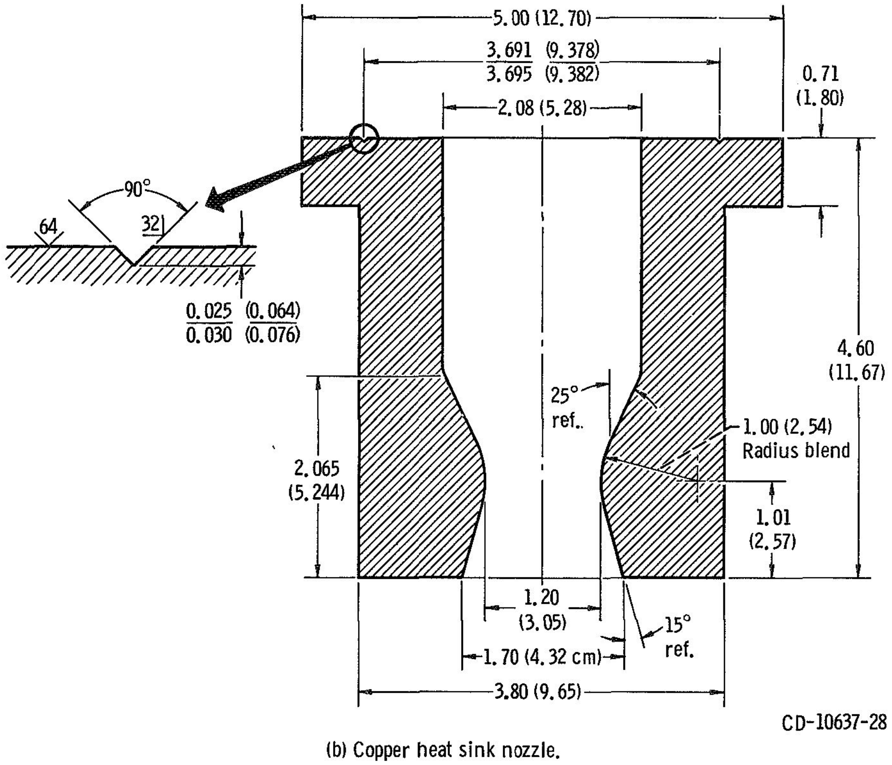
	Axial coolant grooves			
	Depth		Width	
	in.	cm	in.	cm
Chamber	0.100	0.254	0.142	0.361
Throat	.100	.254	.088	.223

(b) Nozzle.

Figure 3. - Water-cooled aluminum chamber and nozzle designed for operation at chamber pressure of 200 psia (1378 kN/m^2). Dimensions are in inches (cm).

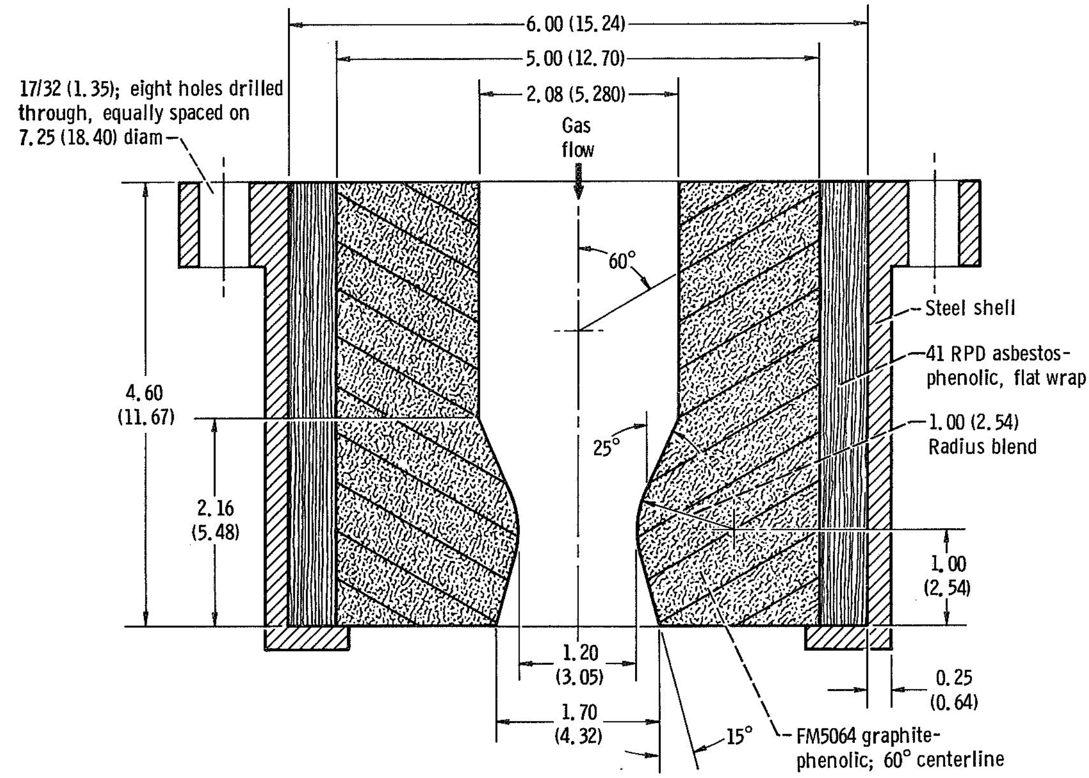


(a) Water-cooled chamber of oxygen-free high-conductivity copper.

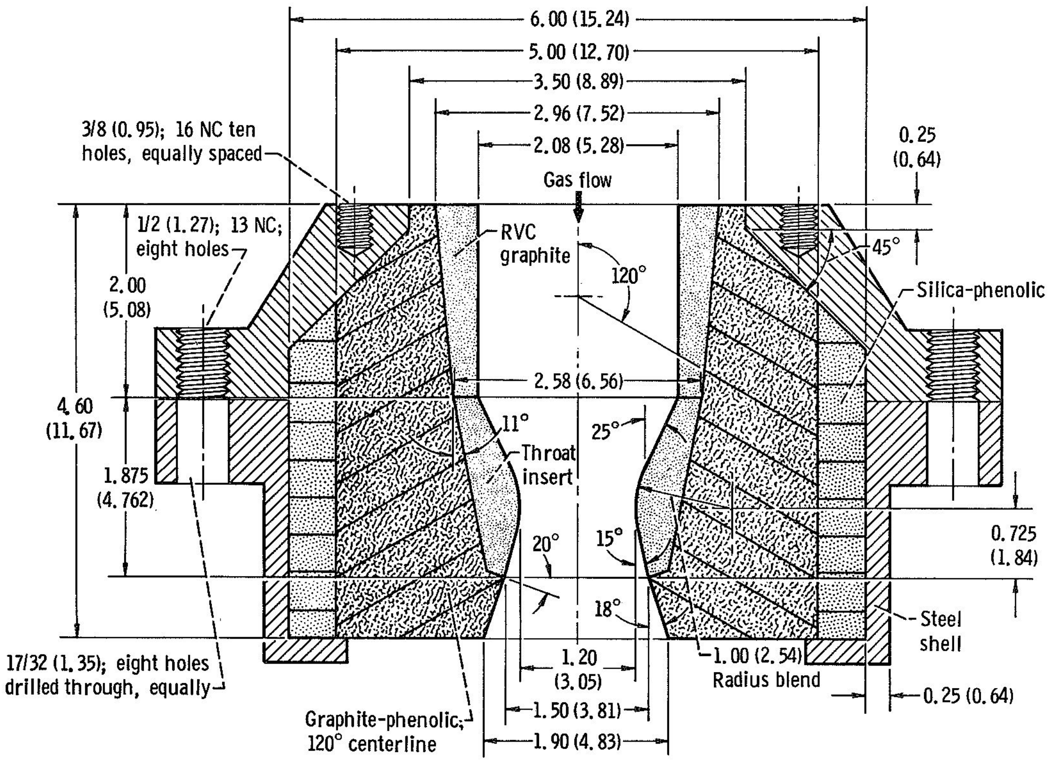


(b) Copper heat sink nozzle.

Figure 4. - Chamber and nozzle designed for operation at chamber pressure of 400 psia (2756 kN/m^2). Dimensions are in inches (cm).



(a) Ablative material.



(b) Ablative material with throat insert.

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Figure 5. - Passively cooled chamber-nozzle design. Dimensions are in inches (cm).

TABLE II. - THROAT INSERT MATERIALS

Class	Name	Supplier
Refractory metals	Tungsten Molybdenum 80%W-20%Cu	General Electric General Electric Firth-Sterling
Composite materials	JTA(SiC, ZrB_2+C) JTO981(ZrC, SiC+C) JTO992(HfC, SiC+C) TaC+C	Union Carbide Union Carbide Union Carbide TRW
Graphites	HPD-1 F50 Carbitex Pyrolytic washers CFZ ATJS	Poco Thiokol Basic Carbon General Electric Union Carbide Union Carbide
Transpirant cooled	Pyrolytic washers (uncooled) Pyrolytic washers (Si_3N_4 cooled)	Marquardt Marquardt

Figure 5(b) shows the chamber-nozzle configuration used to test ablative-throat insert combinations. This design was successfully tested with FLOX-propane at 200 psia (1378 kN/m^2) chamber pressure (ref. 3). Table II lists the various materials evaluated as throat inserts. Each of the materials was selected for its high temperature capability and expected chemical compatibility with FLOX-methane combustion products.

A transpirant-cooled nozzle, designed by the Marquardt Corporation, was also evaluated. A drawing of the nozzle is given on figure 6(a). The concept was to use trisilicon tetranitride (Si_3N_4) as a transpirant coolant in a pyrolytic graphite support matrix. As heat is transferred from the combustion gases to the Si_3N_4, sublimation takes place at approximately 3760° R (2085 K). This reaction absorbs 5600 Btu per pound (13 000 kJ/kg). The vaporized gas is further heated and decomposes to silicon and nitrogen gas in an endothermic reaction which absorbs more heat. The resulting gases are distributed into the combustion gas boundary layer by metering slots shown in figure 6(a). Pyrolytic graphite was selected as the matrix material to provide good radial heat transfer via conduction and radiation from the combustion gases to the transpirant coolant. The coolant phase change could accept a heat flux of 0.7 Btu per square inch per second (1140 kW/m^2) for a 300-second lifetime. The remainder of the heat flux entering the nozzle would have to be absorbed by the endothermic decomposition re-

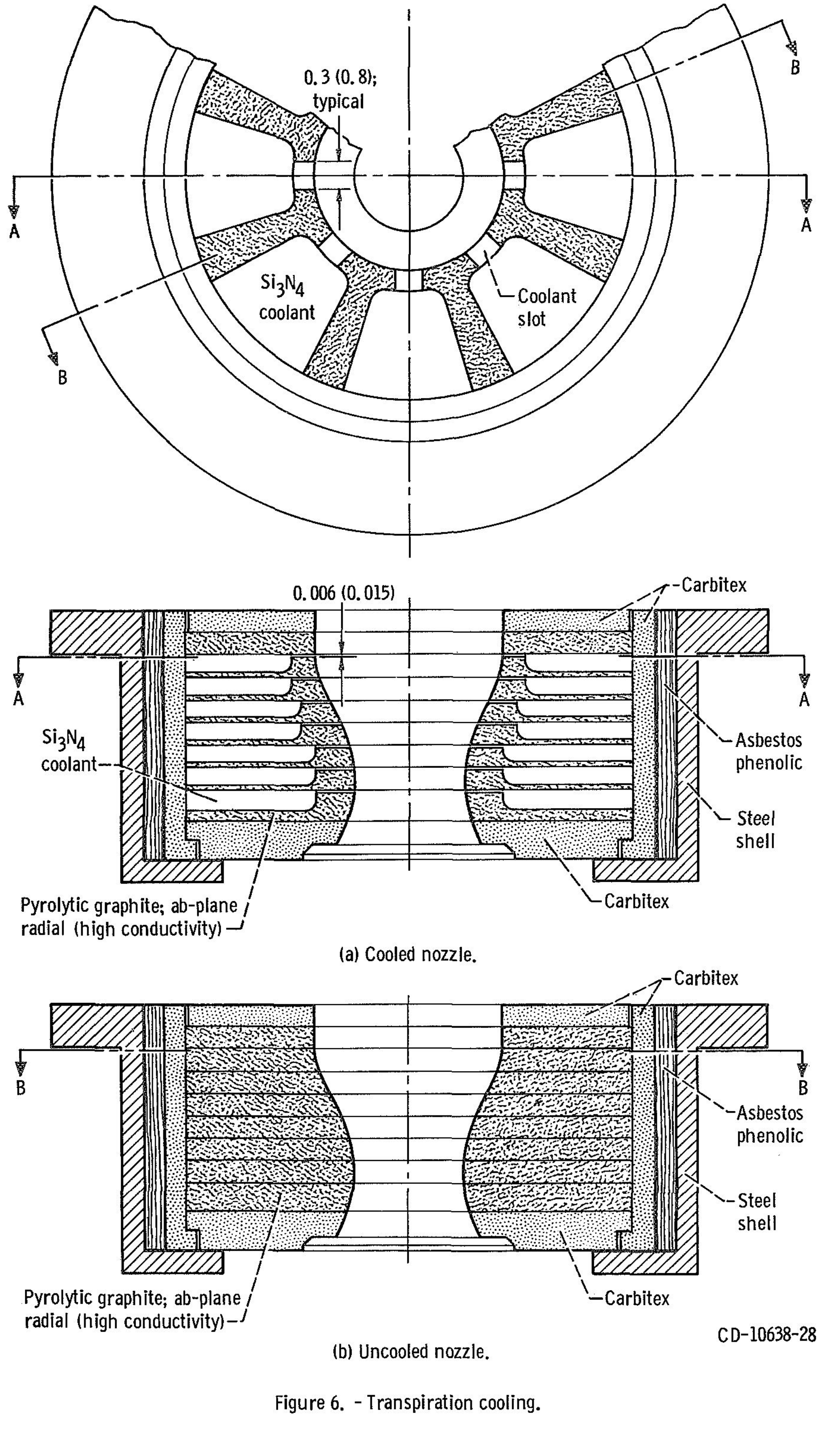


Figure 6. - Transpiration cooling.

action or blocked by film cooling from the gases generated. It was anticipated that the high wall temperature capability of the pyrolytic graphite would enable steady-state operation at heat flux levels below those experienced with water-cooled nozzles (10 Btu/(in.^{2})(sec); 16 300 kW/m^2) and in the range of the coolant capability (3 Btu/(in.^{2})(sec); 4900 kW/m^2).

A control nozzle of solid pyrolytic graphite washers was procured to provide an experimental comparison with uncooled pyrolytic graphite in a similar nozzle configuration. The design is illustrated on figure 6(b).

FACILITY

Test Cell

Test firings were performed in Cell 22 of the Rocket Laboratory Group. The toxic combustion products were condensed or combined with water sprays to prevent atmospheric contamination. The dissolved and condensed products were neutralized in a holding basin before discharge.

Figure 7 is a photograph of the interior of the test facility. The thrust stand with a thrust chamber assembly in place is illustrated. Figure 8 is a schematic of the test facility. The methane tank was surrounded by liquid nitrogen at 100 psia (689 kN/m^2) to

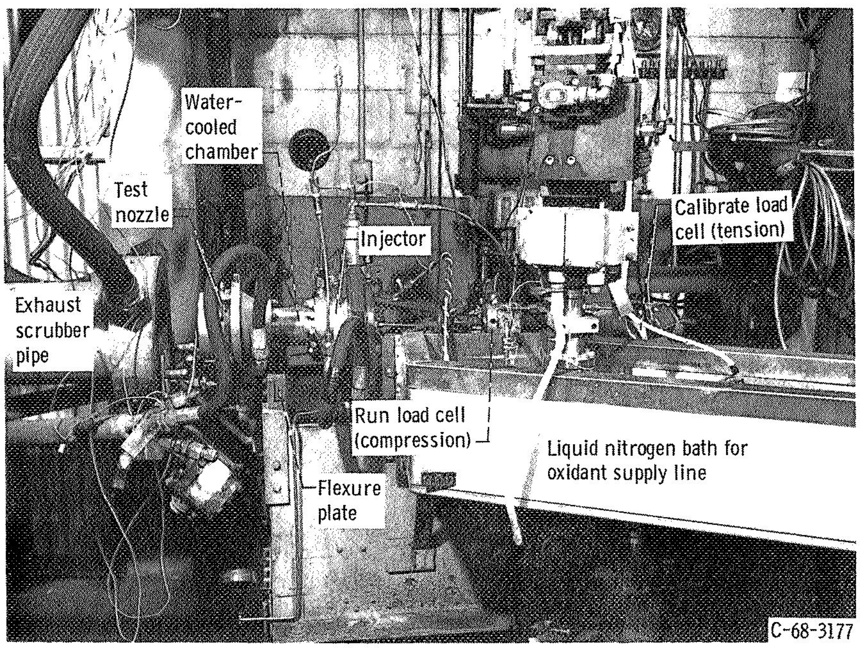


Figure 7. Test facility.

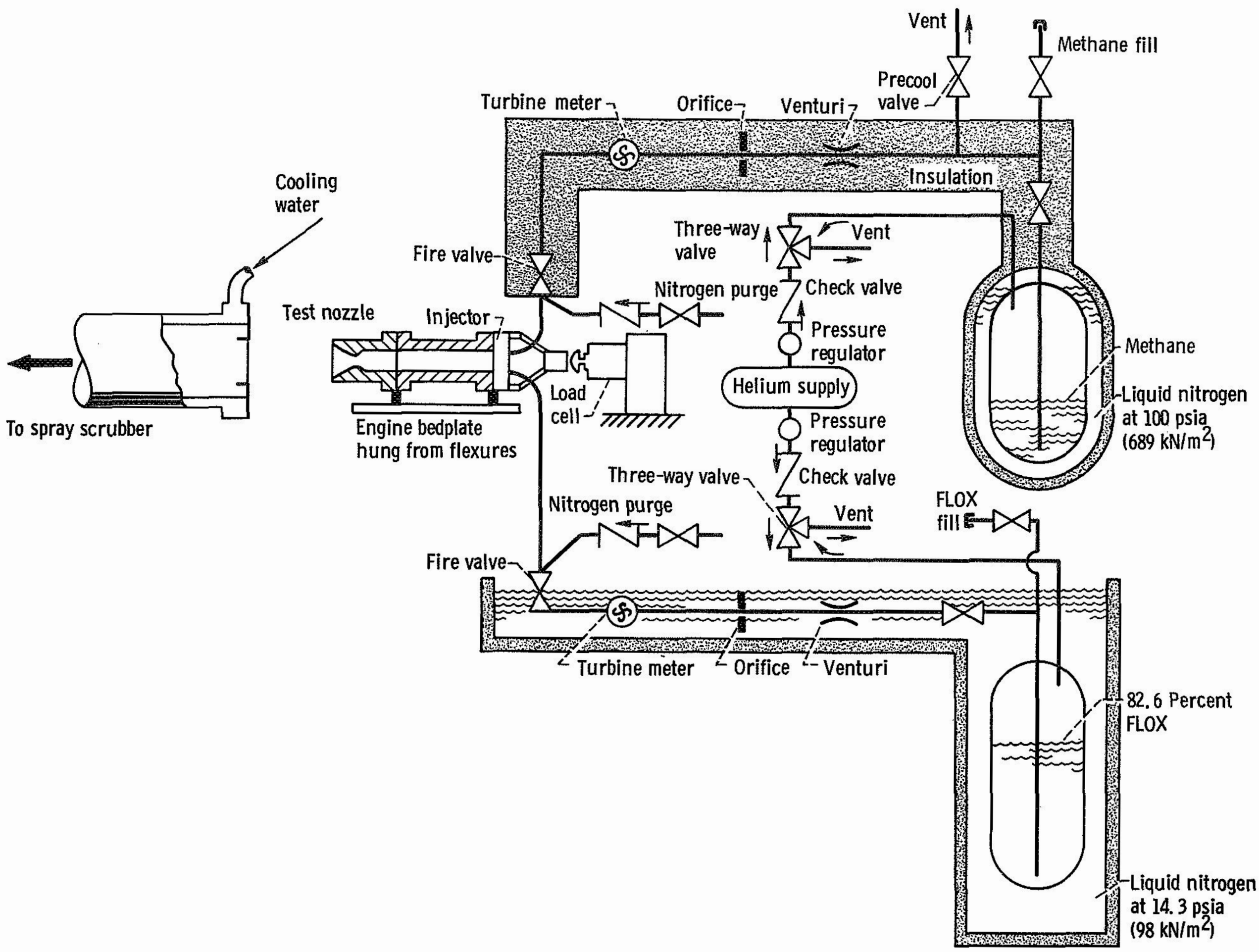


Figure 8. - Schematic diagram of test installation.

keep the temperature of the methane within the liquid range. The actual temperature of the methane was 180° R (100 K).

Instrumentation

Engine chamber pressure was measured by redundant strain-gage-type pressure transducers closely coupled to a hole in the injector face. Thrust measurements were made with a double-bridge strain-gage-type load cell in compression. The load cell was calibrated against a standard load cell.

Flow rates of each propellant were measured by venturi, orifice, and turbine meters in series. The meters were water calibrated and the values were corrected for thermal contraction and propellant density to get actual propellant flows. The data from the three different measuring techniques were averaged to minimize calibration errors due to the meters not being calibrated with the actual propellants.

Propellant temperatures were measured with platinum resistance elements in a bridge circuit. The calibration temperature range was 160° to 240° R (89 to 133 K) for methane and 138° and 148° R (77 to 82 K) for oxidant.

Data Recording and Processing

The transducer outputs were sampled at a commutation rate of 4000 samples per second. For the 80 word block used, each transducer was thus sampled 50 times a second. The sampled outputs were digitized and recorded on magnetic tape. Digital computing equipment was used to convert the data to engineering quantities and perform the necessary calculations.

PROCEDURE

THRUST CHAMBER ASSEMBLY

Figure 9 illustrates a water-cooled thrust chamber assembly. This assembly was used to measure injector performance at 200 psia (1378 kN/m^2) chamber pressure. These test firings were normally 10 to 15 seconds duration. For injector performance testing at 400 psia (2756 kN/m^2), a copper heat sink nozzle was used in place of the water-cooled nozzle. Run duration was between 2 and 4 seconds for these tests.

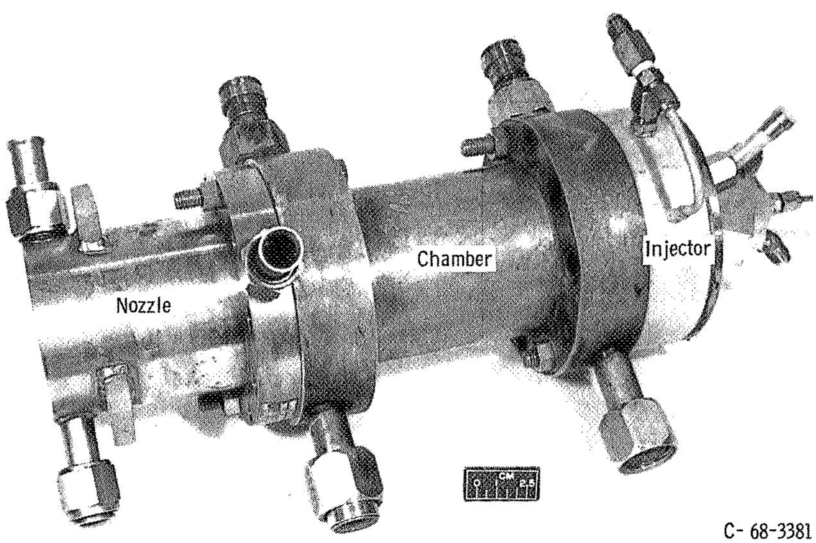


Figure 9. - Water-cooled thrust-chamber assembly.

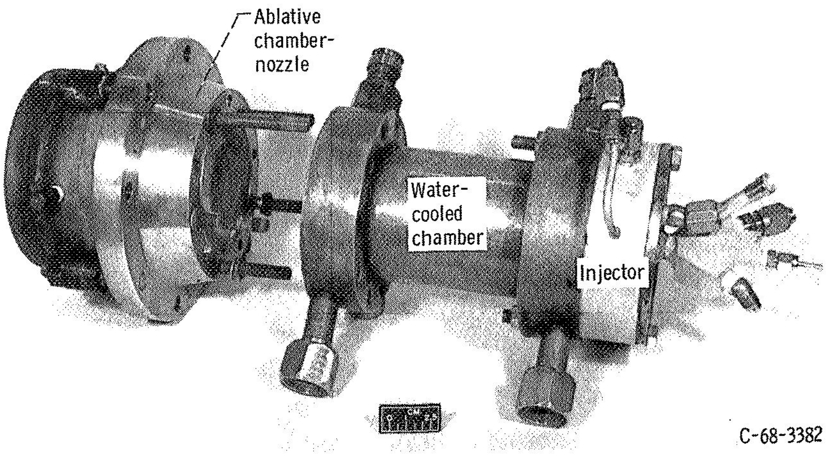


Figure 10. - Ablative chamber-nozzle, water-cooled chamber, and injector.

Figure 10 illustrates the thrust chamber assembly used to test ablative materials and throat inserts. In all of these tests, a water-cooled chamber was included to provide an L* between 31 and 33 inches.

TEST PROCEDURE

Liquid FLOX was mixed in a liquid-nitrogen jacketed weigh tank at a remote site. The liquid fluorine and liquid oxygen were assumed to be completely mixed after being transported by truck trailer to the test site - a distance of 1 mile (1609 m).

The desired value for FLOX was 82.6 weight percent fluorine and 17.4 weight percent oxygen for peak theoretical impulse. The average value obtained by the weigh tank method for 13 different mixes was 82.9 percent fluorine. The precision of the mixing operation was expressed as a standard deviation of ±0.7 percent. Actual values varied between 81 and 84 percent so that for performance calculations, the actual value of the mixture was used.

Precooling of the fuel line before each test was accomplished by running liquid methane through three-quarters of the line length. All instrumentation was electrically calibrated before and after each test firing. An automatic sequence timer activated data acquisition equipment, line purges, and propellant fire valves in the proper sequence. The O/F and chamber pressure were held constant during each test firing by separate closed loop controller systems which adjusted the appropriate propellant flow rates. Selected transducer outputs such as chamber pressure and propellant flow rates were recorded on oscillograph and strip charts in the control room. These data were used for system performance checks and to decide upon test termination if excessive erosion or other problems arose.

The three flowmeters used to measure oxidant weight flow agreed within ±1.0 percent of one another. The three fuel flowmeters agreed within ±4.0 percent of one another. Since no obvious method of resolving the discrepancies was apparent, the average of all three meters was used for each propellant flow.

CALCULATIONS

Pertinent calculations along with the definition of symbols are listed in the appendix. After each test of an ablative or throat insert material, an image of the throat plane as projected by a 10X optical comparator was traced. A planimeter was used to measure the nozzle throat area and the area was converted to an effective throat radius.

RESULTS AND DISCUSSION

INJECTOR EVALUATION

The characteristic exhaust velocity efficiency η_{C^*} was calculated as indicated in the appendix. The precision of the C* efficiency was ±0.8 percent for one standard deviation. The precision was determined by a propagation of errors calculation as well as from the reproducibility of duplicate measurements. No corrections were made for engine friction losses, throat shrinkage due to thermal stresses, or heat losses to the thrust chamber wall.

A summary of measured injector performance is given in table III. The perform-

TABLE III. - MEASURED INJECTOR PERFORMANCE

Injector	Chamber pressure, P_c		Injector pressure drop of -				Characteristic velocity efficiency (O/F = 5.5), $\eta_{C^*_{exp}}$, percent	Characteristic chamber length, L*	
			Oxidant, ΔP_{ox}		Fuel, ΔP_f				
	psia	kN/m^2	psi	kN/m^2	psi	kN/m^2		in.	cm
1	200	1378	219	1509	153	1054	96.5	33	83.8
2	400	2756	256	1764	208	1433	98.4	31	78.7
3	400	2756	183	1261	123	847	98.4	31	78.7

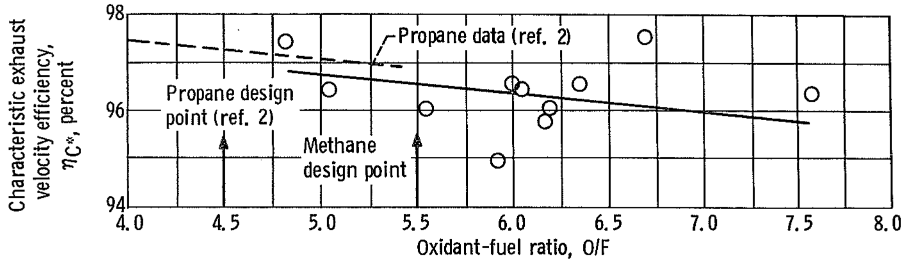


Figure 11. - Performance of injector 1 (37-element triplet) at chamber pressure of 200 psia (1378 kN/m^2). Water-cooled thrust chamber; propellant, FLOX (83 percent fluorine and 17 percent oxygen) - methane; throat diameter, 1.2 inches (3.05 cm); characteristic chamber length, 33 inches (83.8 cm).

ance of injector 1 at 200 psia (1378 kN/m^2) chamber pressure is plotted on figure 11. The value of η_{C^*} = 96.5 percent at the design O/F of 5.5 was satisfactory. This compares with an η_{C^*} of 97.2 percent for FLOX-propane at an O/F of 4.5 for a similar injector.

Figure 12 gives the η_{C^*} of injectors 2 and 3 at 400 psia (2756 kN/m^2) chamber pressure. Both injectors were about 98.4 percent efficient at the design point. Injector 3 had increased efficiency at the lower O/F's, however. The reason for this was probably better mixing provided by the added elements of injector 3. The added elements were arranged radially to improve propellant distribution for better ablative compatibility. Previous work of reference 3 indicated ablative performance could be improved by operating at lower O/F values. Since operation at lower O/F's could involve performance penalties, injector 3 was selected for ablative and throat insert evaluation to minimize these penalties. The vacuum specific impulse was predicted for an engine

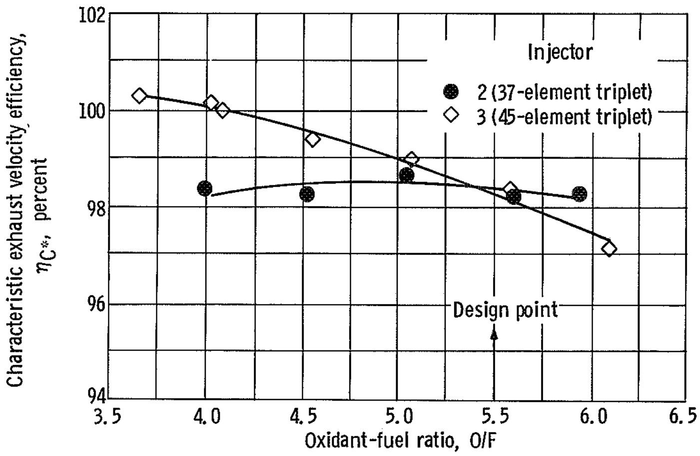


Figure 12. - Performance of injectors 2 and 3 at chamber pressure of 400 psia (2756 kN/m^2). Water-cooled thrust chamber and heat sink nozzle; propellant, FLOX (82.6 percent fluorine and 17.4 percent oxygen) - methane; throat diameter, 1.2 inches (3.05 cm); characteristic chamber length, 31 inches (78.8 cm).

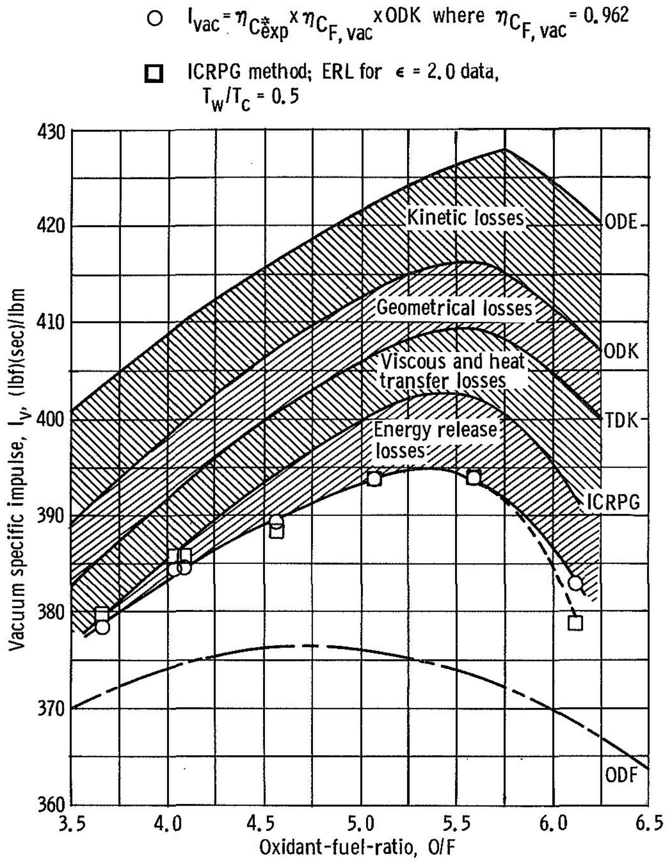


Figure 13. - Predicted engine performance. Injector 3; 82.6 percent fluorine; expansion area ratio, 60; throat diameter, 1.2 inches (3.05 cm); chamber pressure, 400 psia (2756 kN/m²); conical nozzle, $\alpha = 15°$.

using injector 3 and a nozzle with an expansion area ratio of 60:1 (see fig. 13). Two calculation procedures were used: the ICRPG method of reference 6 and a constant vacuum thrust coefficient efficiency method. Both procedures are based upon low area ratio nozzle tests and attempt to calculate all additional engine losses for extrapolation to high area ratio. The ICRPG method uses low area ratio tests to calculate energy release losses (ERL) upon which to base the extrapolation. The constant $\eta_{C_{F,vac}}$ method makes modifications to $\eta_{C^*_{exp}}$ using calculated $\eta_{C_{F,vac}}$ (ref. 7) and laminar stagnation point heat transfer correlation (C = 0.5) of reference 8. Both methods give virtually the same predicted value of 395 seconds peak impulse at an O/F of 5.3. Ideally, one would operate this engine at the O/F for maximum delivered specific impulse (5.3). If ablative compatibility requires operation at O/F lower than 5.3, performance would have to be compromised. It might be possible to attain high performance with good ablative compatibility through the use of O/F zoning of the injector. However, the scope of this program did not include an exploration of zoned injection.

CHAMBER EVALUATION

The injector evaluation work was performed with complete water-cooled thrust chambers at 200 psia (1378 kN/m^2) chamber pressure and with a water-cooled chamber section at 400 psia (2756 kN/m^2).

The water-cooled chamber shown in figure 4(a) (p. 7) was originally designed to operate satisfactorily at a heat flux level of 6 Btu per square inch per second (9800 kW/m^2). Operation at 200 psia (1378 kN/m^2) chamber pressure produced a heat flux level between 3 and 4 Btu per square inch per second (4900 and 6525 kW/m^2). Operation at 400 psia (2756 kN/m^2) chamber pressure produced heat flux levels from 7 to 10 Btu per square inch per second (11 400 to 16 300 kW/m^2). The higher values exceeded the cooling capacity in some cases and resulted in burnout.

A water-cooled chamber was therefore designed as shown in figure 5(a) (p. 8) to operate satisfactorily at a heat flux level of 10 per square inch per second (16 300 kW/m^2). Tests at 400 psia (2756 kN/m^2) chamber pressure normally resulted in heat flux levels from 4 to 8 Btu per square inch per second (6525 to 13 050 kW/m^2). The observed carbon deposition attenuated the heat flux somewhat.

ABLATIVE MATERIALS EVALUATION - EFFECT OF CHAMBER PRESSURE AND O/F

Figure 14 illustrates the effect of O/F variation on ablative throat erosion at 200 psia (1378 kN/m^2) chamber pressure. The results are similar to those reported

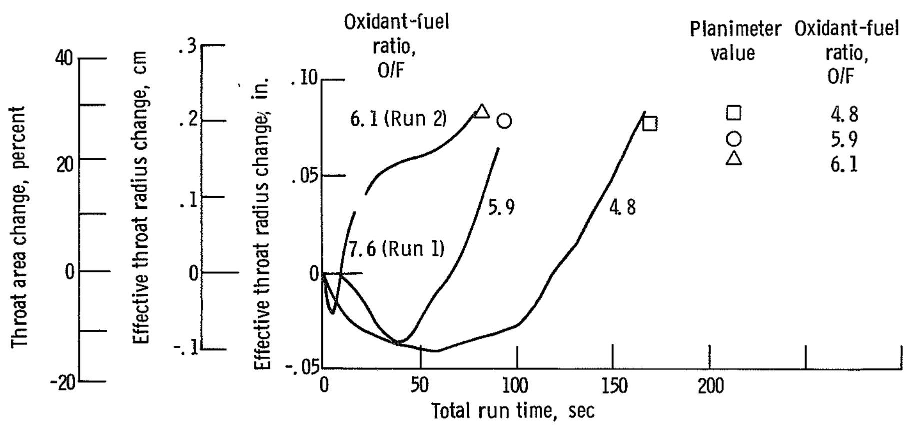


Figure 14. - Effect of oxidant-fuel ratio on ablative erosion at chamber pressure of 200 psia (1378 kN/m^2). Graphite-phenolic, 60° centerline; propellant, FLOX-methane; throat diameter, 1.2 inches (3.05 cm).

for FLOX-propane (ref. 3) in that lower O/F values result in an initial throat area decrease caused by carbon and pyrolytic graphite deposition. The pyrolytic graphite, deposited as insulating layer planes, protects the ablative initially. As the run continues the temperature increases and the protective layer is removed. Finally, loss of the ablative material occurs due to oxidation and stream shear forces. For the material and injector combination tested, the throat area increase exceeded 15 percent after approximately 150 seconds at an O/F of 4.8. The open symbols are post-test measurements made by a planimeter.

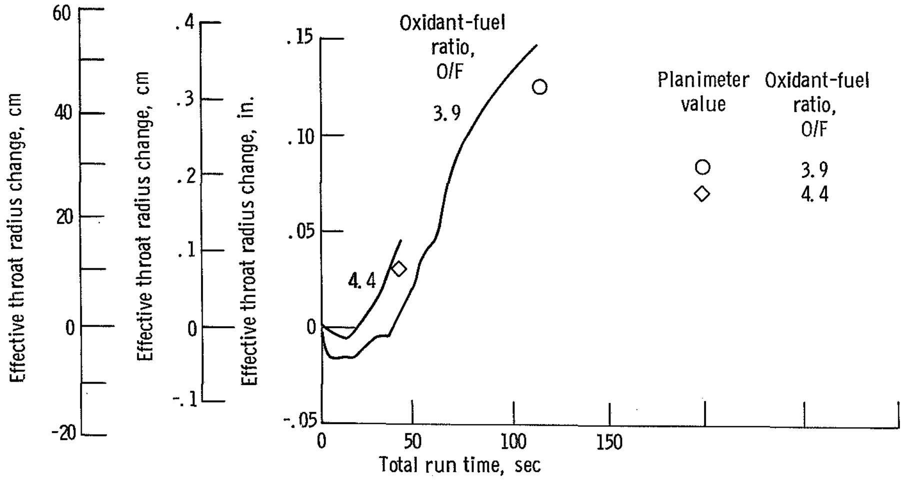


Figure 15. - Effect of oxidant-fuel ratio on ablative erosion at chamber pressure of 400 psia (2756 kN/m²). Graphite-phenolic, 60° centerline; propellant, FLOX-methane; throat diameter, 1.2 inches (3.05 cm).

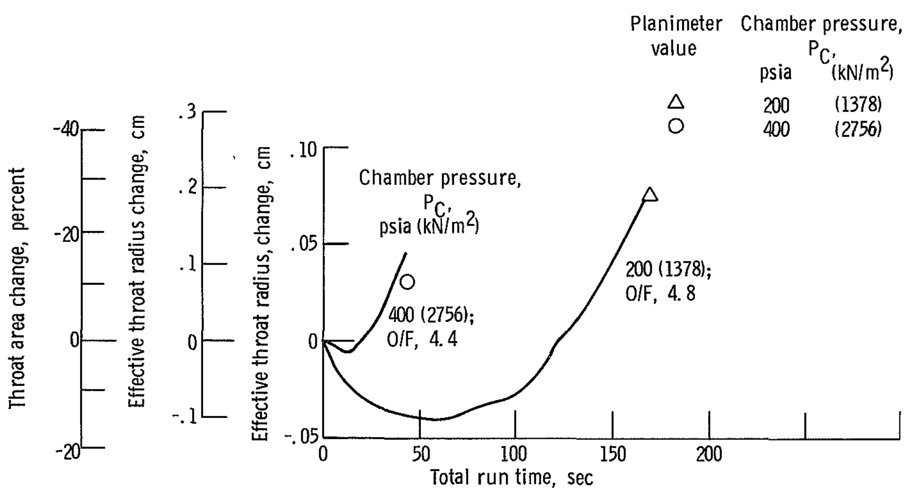
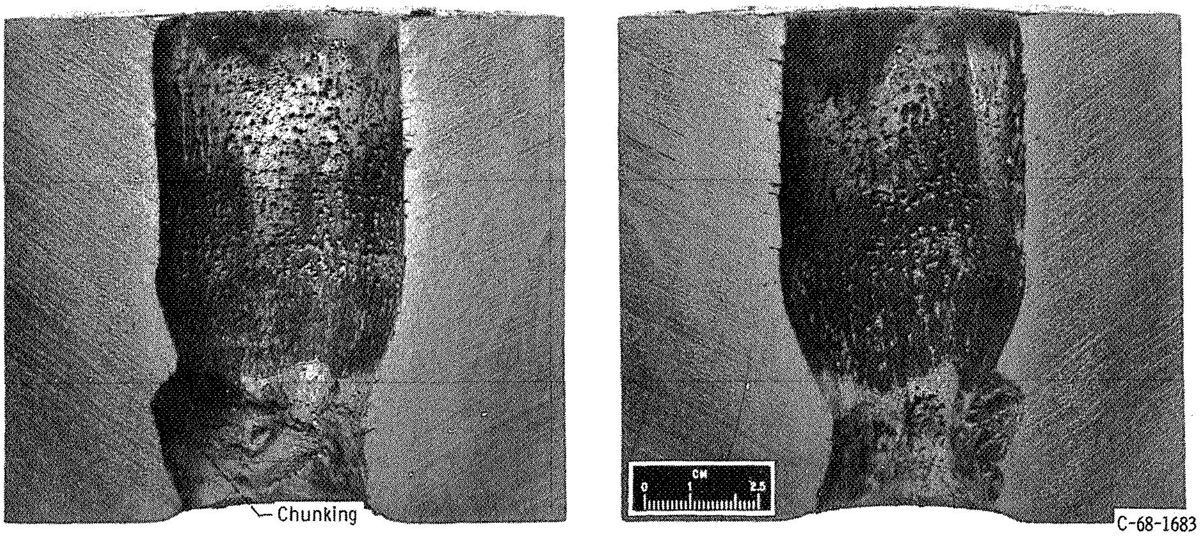
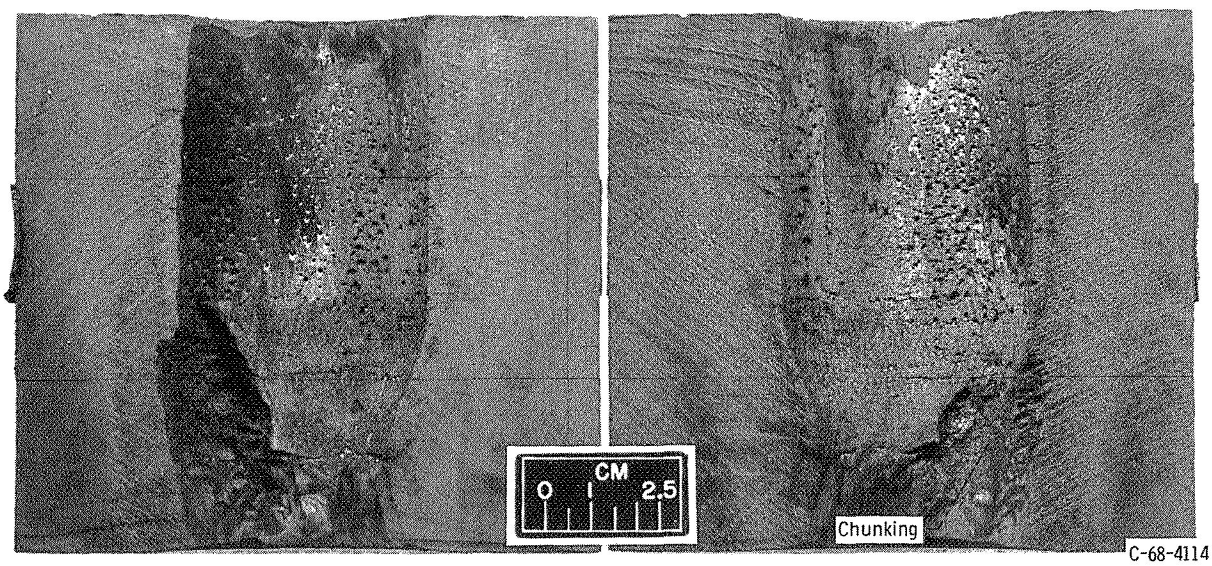


Figure 16. - Effect of chamber pressure on ablative erosion. Graphite-phenolic, 60° centerline; propellant, FLOX-methane; throat diameter, 1.20 inches (3.05 cm).

When the chamber pressure increased to 400 psia (2756 kN/m^2), the throat erosion increased sharply, as can be seen from figure 15. Even when operating at an O/F value as low as 3.9, the time required for a throat area increase of 15 percent was less than 60 seconds. Figure 16 is presented to clearly illustrate the dependency of ablative material life on chamber pressure. Both runs were made at approximately the same O/F (4.4 and 4.8). Operation at 400 psia (2756 kN/m^2) decreases the useful lifetime to 40 seconds compared with approximately 150 seconds at 200 psia (1378 kN/m^2). The



(a) Chamber pressure, 200 psia (1378 kN/m^2); total firing time, 170 seconds; oxidant-fuel ratio, 4.8.



(b) Chamber pressure, 400 psia (2756 kN/m^2); total firng time, 43 seconds; oxidant-fuel ratio, 4.4.

Figure 17. - Graphite-phenolic chamber nozzle, 60° centerline. Propellant, FLOX-methane.

protective pyrolytic graphite coating is removed much sooner at 400 psia (2756 kN/m^2) due to the increased heat-transfer coefficient, with corresponding higher oxidation rates. After removal of the coating, the ablative erodes rapidly due to oxidation and the high shear forces. The difference in erosion performance would have been greater had the O/F's been the same. Inspection of the tested nozzles (fig. 17) illustrates the failure mechanism. Severe material chunking is evident at 400 psia (2756 kN/m^2) chamber pressure. A nonuniform erosion pattern can also be seen.

From the results of the preceding testing, it was obvious that the value of standard ablative materials is limited at the desired operating conditions. Three possible solutions to the ablative erosion problem are possible. One would be to develop an injector which produced a more uniform combustion environment. Another solution might be to zone the injector by O/F distribution to protect the ablative from oxidation and high temperature. A third method to combat the ablative throat erosion problem would be the installation of a throat insert to prevent shear-oxidation failures. Although a combination of all three methods would be most desirable, the throat insert approach was the only one evaluated in this program.

THROAT INSERT EVALUATION

The first step was to determine what O/F value was required to minimize throat area changes when the thrust chamber was operated at 400 psia (2756 kN/m^2) chamber pressures. Too high an O/F would cause oxidation failure. Too low an O/F would cause excessive carbon deposition. An O/F near the maximum delivered impulse point (5.2 - 5.3) would be ideal for performance.

O/F Effect

To determine the O/F required for minimum throat area change, a high density, isotropic graphite (ATJS) was used as a throat insert (see fig. 5, p. 8). Figure 18 shows the erosion data for mixture ratios of 4.1 and 4.5. Higher O/F values were not run due to hardware limitations. Operation at 400 psia (2756 kN/m^2) chamber pressure and an O/F of 4.1 was found to minimize throat area changes and the erosion free run duration was also appreciably longer than was possible with ablatives (see fig. 15). Since an O/F of 4.1 was below the O/F of 5.3 required for maximum delivered specific impulse, design of an optimum thrust chamber assembly would require some compromise with performance. The most likely method would be to zone the O/F for 4.1 at the wall with compensating higher O/F in the core in an attempt to achieve maximum impulse performance.

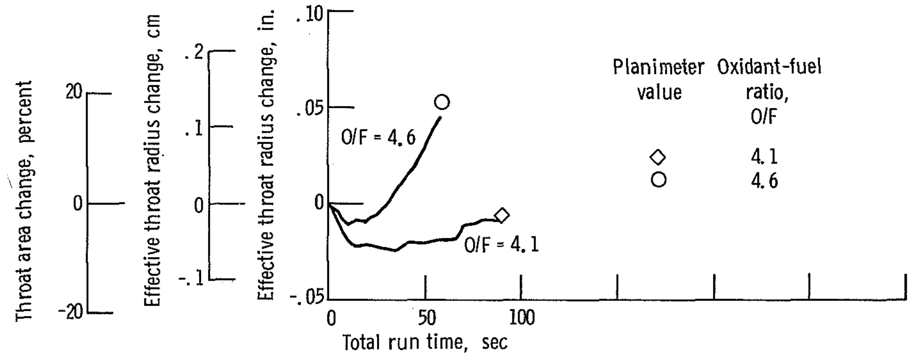


Figure 18. - Effect of oxidant-fuel ratio on throat insert erosion. ATJS graphite; propellant, FLOX-methane; throat diameter, 1.2 inches (3.05 cm); chamber pressure, 400 psia (2756 kN/m^2).

Materials Evaluation (400 psia (2756 kN/m^2); O/F, 4.1)

To determine the general suitability of various refractory materials on the FLOX-methane environment, the following throat insert materials were evaluated. No attempt was made to optimize the material design concepts at this time but only to screen classes for potential usefulness.

Refractory metals. - Refractory metals tested at 400 psia (2756 kN/m^2) chamber pressure and an O/F of 4.1 gave the results shown on figure 19. Visual post-test examination revealed that the rapid erosion was due to melting of the molybdenum and oxidation of the tungsten and copper impregnated tungsten. A significant decrease in the O/F and/or surface temperature would be required to utilize these materials. The development of a suitable coating system may help.

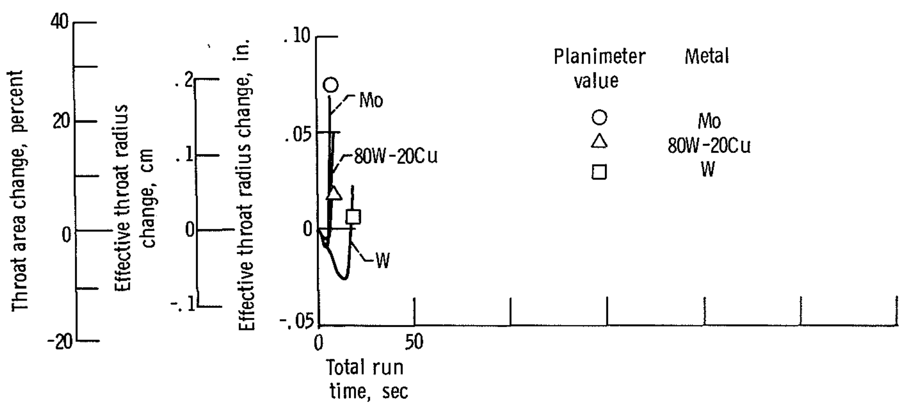


Figure 19. - Throat insert material evaluation of refractory metals. Propellant, FLOX-methane; chamber pressure, 400 psia (2756 kN/m^2); throat diameter, 1.2 inches (3.05 cm); oxidant-fuel ratio, 4.1 to 4.5.

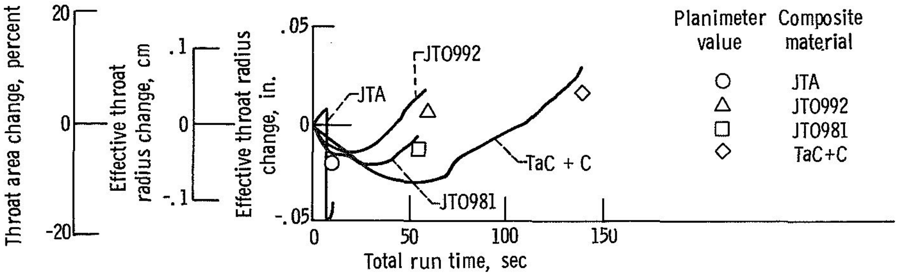
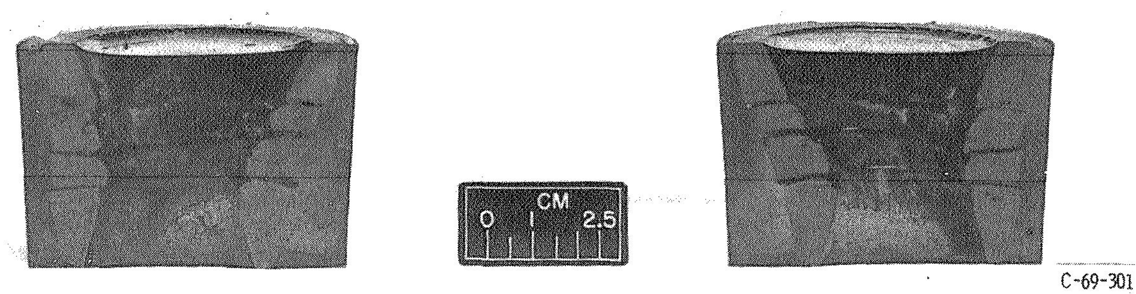
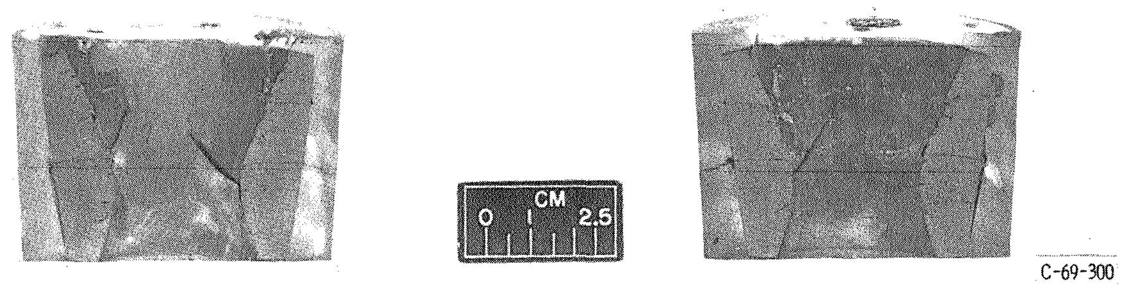


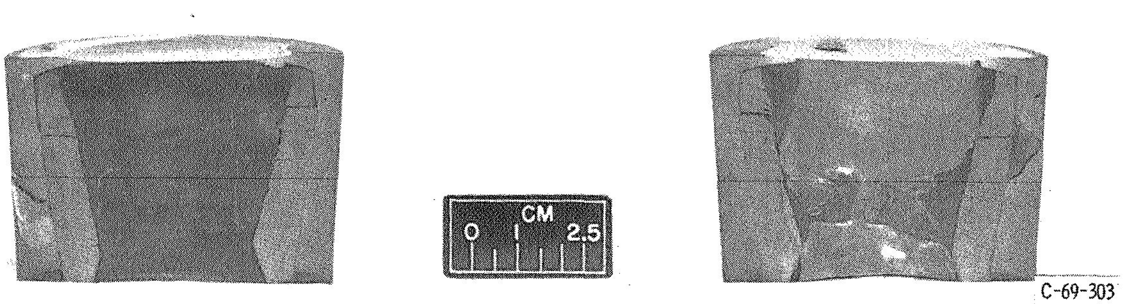
Figure 20. - Throat insert material evaluation of composite materials. Propellant, FLOX-methane; chamber pressure, 400 psia (2756 kN/m^2); throat diameter, 1.2 inches (3.05 cm); oxidant-fuel ratio, 4.0 to 4.2.



(a) Material, JTA; firing time, 10 seconds.



(b) Material, JTO981; firing time, 55 seconds.



(c) Material, JTO992; firing time, 60 seconds.

Figure 21. - Throat inserts after test. Chamber pressure, 400 psia (2756 kN/m^2); propellant, FLOX-methane; oxidant-fuel ratio, 4.0 to 4.2; throat diameter, 1.2 inches (3.05 cm).

Composite materials. - Figure 20 gives the erosion results for some composite materials. All of these materials failed structurally by thermal stress. Except for the 10-second run of JTA graphite, all of the composites showed signs of oxidation or vaporization. The structural failures of three materials are illustrated in figure 21. The fourth material and the best of the composites tested was a TaC plus 30 volume percent graphite. Throat erosion due to structural failure and oxidation began after 135 seconds. The post-test condition of the insert is shown on figure 22. A possible solution to the

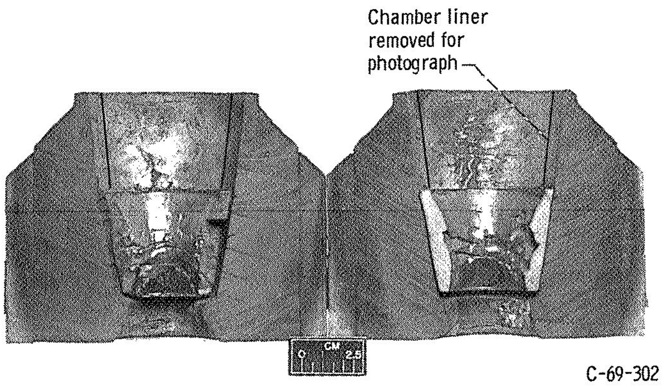


Figure 22. - TaC + C insert after test. Chamber pressure, 400 psia (2756 kN/m^2); propellant, FLOX-methane; oxidant-fuel ratio, 4.16; throat diamter, 1.2 inches (3.05 cm); total firing time, 140 seconds.

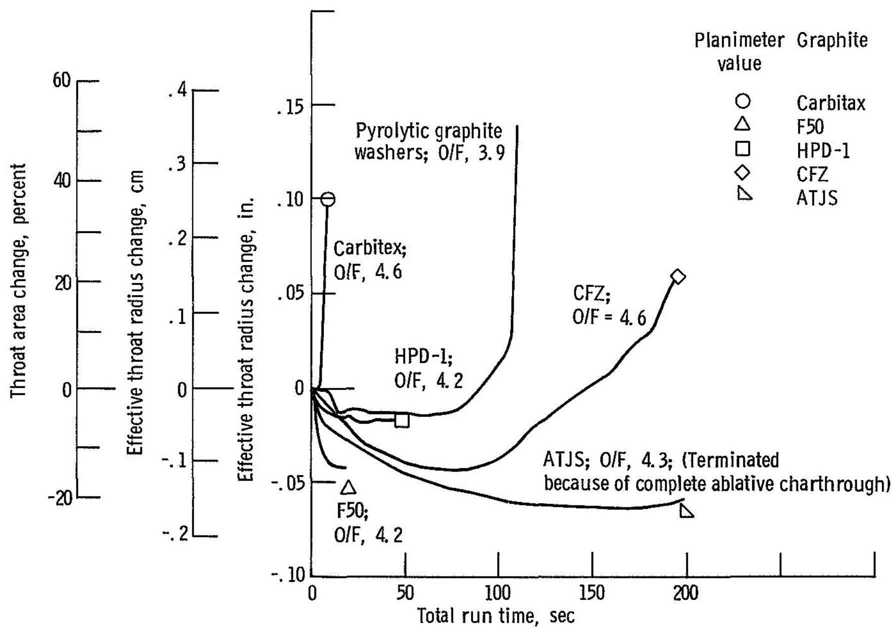


Figure 23. - Evaluation of graphites as throat insert materials. Propellant, FLOX-methane; chamber pressure, 400 psia (2756 kN/m^2); throat diameter, 1.2 inches (3.05 cm).

structural failure problem may be the addition of more graphite, which would lower the modulus of elasticity of the composite.

Graphites. - Figure 23 presents the erosion results for graphite materials. The F50 material was a castable compound intended for economy of production. The material failed structurally, as illustrated in figure 24, and allowed a combustion leak which terminated testing. The HPD-1 material was also a low cost material which failed structurally (cracks occurred during firing) as illustrated in figure 25. The Carbitex and pyrolytic graphite inserts were both constructed from axial segments (washers) with

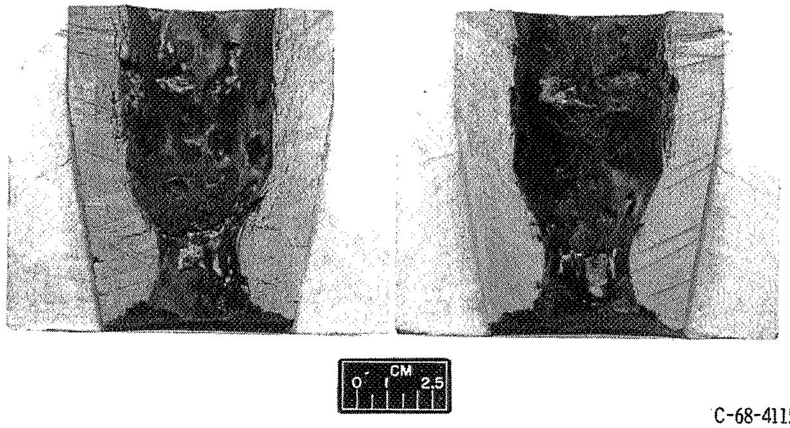


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Figure 24. - F50 molding compound chamber nozzle. Chamber pressure, 400 psia (2756 kN/m^2); propellant, FLOX-methane; oxidant-fuel ratio, 4.2; total firing time, 20 seconds.

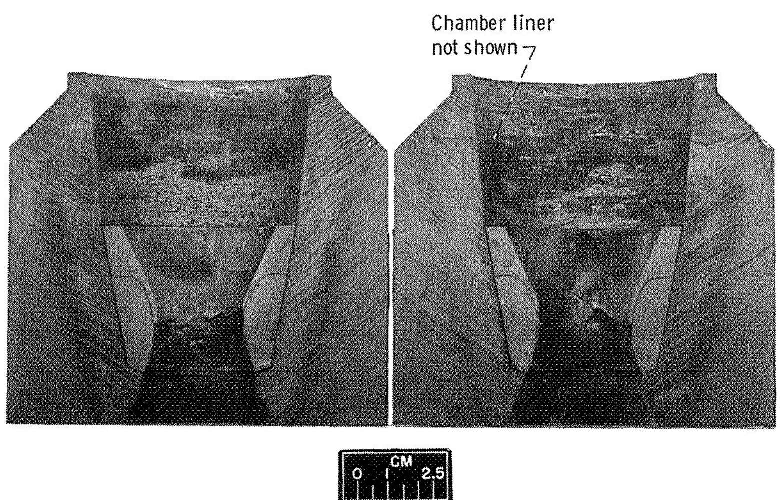

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Figure 25. - HPD-1 graphite throat insert. Chamber pressure, 400 psia (2756 kN/m^2); propellant, FLOX-methane; oxidant-fuel ratio, 4.2; total firng time, 40 seconds.

high conductivity in the radial direction. In both cases, the layers failed structurally and were completely ejected from the throat, causing the steep erosion curves of figure 23.

The insert made of CFZ graphite was moderately successful. Figure 26 illustrates the condition of the CFZ insert after the test firing. A circumferential crack is evident, and the surface is rough and grainy. The erosion mode seems to be largely loss of particles due to a combination of oxidation and stream shear forces.

The best insert material tested was ATJS graphite. No throat erosion or structural failure was found after 205 seconds continuous firing. Figure 27 illustrates the complete ablative char-through, which necessitated termination, and the insert erosion upstream

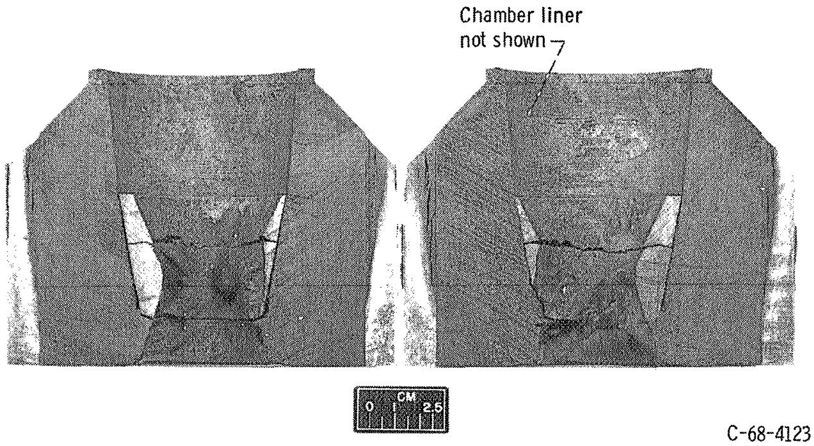


Figure 26. - CFZ graphite throat insert. Chamber pressure, 400 psia (2756 kN/m^2); propellant, FLOX-methane; oxidant-fuel ratio, 4.6; total firing time, 195 seconds.

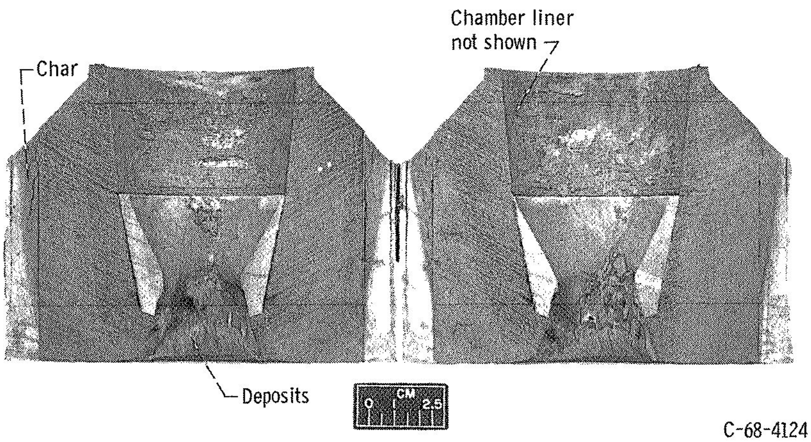


Figure 27. - ATJS graphite throat insert. Chamber pressure, 400 psia (2756 kN/m^2); propellant, FLOX- methane; oxidant-fuel ratio, 4.3; total firing time, 205 seconds.

of the throat. The erosion upstream may have been due to the nonuniform injection pattern which could possibly be corrected by an improved injector manifold design. Also note the graphite material deposition aft of the throat.

Transpiration cooling. - Testing to determine the effectiveness of the transpirant cooled Marquardt design began with the uncooled nozzle (the solid lines of fig. 28). The first firing at an O/F of 4.2 ended after 71 seconds because of a combustion gas leak. As the throat erosion curve shows, carbon deposition caused a decrease in throat area.

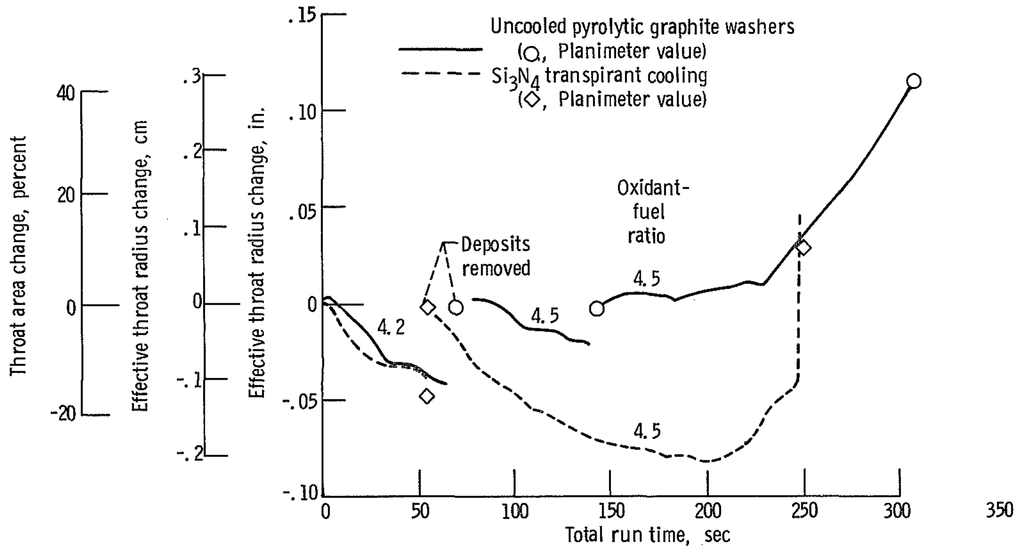
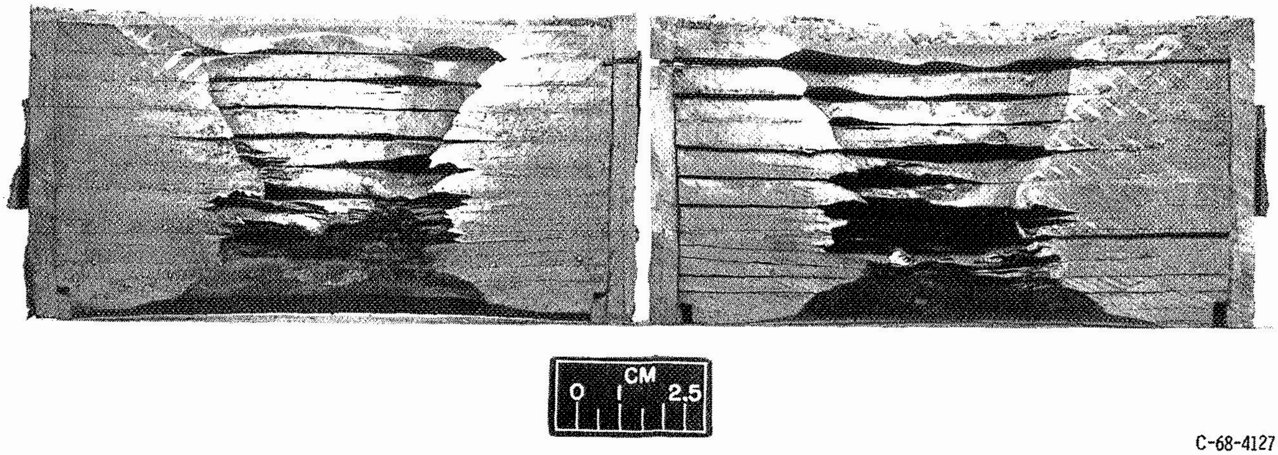


Figure 28. - Evaluation of transpirant cooling concept for throat insert materials. Chamber pressure, 400 psia (2756 kN/m^2); throat diameter, 1.2 inches (3.05 cm).

The deposits were a hard dense material but loosely bonded to the pyrolytic graphite washers. The deposits were removed prior to the next firing. Based on the rate of carbon deposits of the first firing, it was decided to make the next firing at an O/F of 4.5. The second firing was aborted after 68 seconds because of a combustion gas leak. No deterioration of the nozzle was noted and the carbon deposits were minimal. A third firing (O/F = 4.5) was ended after 166 seconds because of erosion. Inspection of the nozzle after a total of 305 seconds total firing time indicated some oxidation but mainly delamination and shear failure (see fig. 29(a)).

Testing of the Si_3N_4 cooled nozzle began at an O/F of 4.5 and was terminated after 55 seconds because of a fuel leak in the feed system. Inspection of the nozzle revealed loose scaly deposits and no throat erosion. X-rays and weight values indicated 20 percent of the coolant Si_3N_4 had been used. For the same rate of coolant usage, the total coolant would be used in 275 seconds. The coolant was also calculated to be absorbing heat at a rate which decreased the wall temperature 150° R (83 K) without any film cool-



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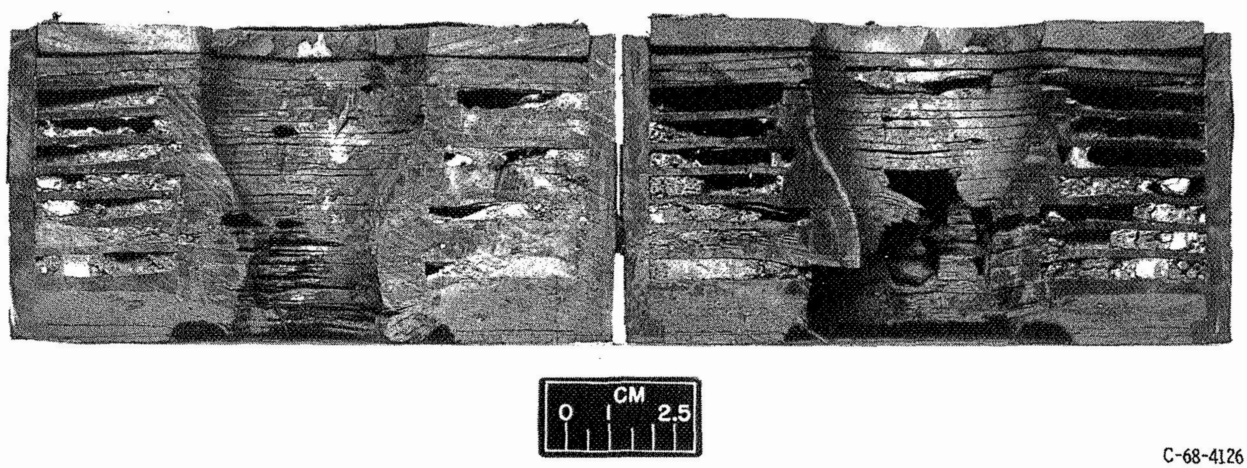
(a) Uncooled. Total firing time, 308 seconds.



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(b) Cooled with Si_3N_4. Total firing time, 305 seconds.

Figure 29. - Pyrolytic graphite washer chamber nozzle. Chamber pressure, 400 psia (2756 kN/m^2); propellant, FLOX-methane; oxidant-fuel ratio, 4.5.

ing effect. Since the nozzle seemed capable of meeting the design lifetime (300 sec total), it was decided to make another longer firing.

The second firing was terminated after 193 seconds because of rapid throat erosion. Although throat erosion was not excessive, the rate shown on figure 28 indicates catastrophic failure. As shown in figure 29(b), delamination and loss of material was severe. The primary failure mode seemed to be structural. Coolant depletion was more pronounced in the chamber section than at the throat plane. This was most likely because the surface to volume ratio in the chamber was three times that of the throat. This effect was apparently more than enough to offset the higher heat flux and lower static pressure at the throat plane. In view of the performance of the cooled and uncooled nozzles, the cooled design did not seem to enhance the performance enough to make the added cost and complexity worthwhile.

SUMMARY OF RESULTS

The results of a program to design a FLOX-methane thrust chamber assembly for operation up to 400 psia (2756 kN/m^2) chamber pressure are summarized as follows:

1. A 45-element, oxidant-on-fuel triplet injector was designed for operation at 400 psia (2756 kN/m^2) chamber pressure. A characteristic velocity efficiency $\eta_{C^*_{exp}}$ of 98.4 percent was obtained at the design O/F of 5.5.

2. Based on $\eta_{C^*_{exp}}$ profiles, peak delivered vacuum specific impulse of 395 seconds (ϵ = 60) was calculated to occur at 5.3 O/F, while the theoretical kinetic peak is an O/F of 5.5.

3. An O/F of approximately 4.1 was required to minimize throat insert area changes due to oxidation or carbon deposition for long duration runs at 400 psia (2756 kN/m^2) chamber pressure.

4. Deposition rates of carbon and pyrolytic graphite increase as a function of decreasing local O/F ratio and proceed until the temperature exceeds some critical value.

5. Graphite phenolic ablative materials were satisfactory at the rocket throat for about 150 seconds at chamber pressures of 200 psia (1378 kN/m^2) with the injectors designed herein.

6. Monolithic graphites, particularly ATJS, were found to be most suitable for throat inserts in the FLOX-methane combustion environment as determined by a 200 second continuous firing at 400 psi (2756 kN/m^2) chamber pressure.

7. Refractory metals were found to be unsuitable in the FLOX-methane environment at 400 psia (2756 kN/m^2) chamber pressure. Composite carbides require further research to determine optimum compositions.

8. A transpiration-cooling concept resulted in only a slight improvement in the performance of an uncooled pyrolytic graphite washer design after approximately 200 seconds firing duration at 400 psi (2756 kN/m^2) chamber pressure. Both concepts failed structurally, however.

9. The normal heat flux to the water cooled combustion chamber and nozzle was experimentally determined to be 4 to 8 Btu per square inch per second (6525 to 13 050 kW/m^2) at a chamber pressure of 400 psia (2756 kN/m^2).

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, October 30, 1969,
128-31.

APPENDIX - SYMBOLS

A_e	rocket nozzle exit area, in.^{2}; cm^2
A_f	fuel injection area, ft^2; m^2
A_{ox}	oxidant injection area, ft^2; m^2
A_t	rocket nozzle throat area, in.^{2}; cm^2
C	heat transfer correlation constant
C_d	nozzle discharge coefficient (0.985)
C_F	thrust coefficient, $F/P_{c,corr}A_t$
$C_{F,d}$	design thrust coefficient (shifting equilibrium)
C_m	momentum coefficient $(Cd)^2$
C^*_{theor}	theoretical characteristic exhaust velocity (shifting equilibrium), ft/sec; m/sec
F	thrust, lbf; N
F_{vac}	vacuum thrust, $F + P_0A_e$, lbf; N
g	gravitational constant, 32.174 ft/sec^2; 9.8 m/sec^2
I_{vac}	vacuum specific impulse, F_{vac}/W_p; sec
$I_{vac,theor,equil}$	theoretical equilibrium vacuum specific impulse
L*	characteristic chamber length, V_c/A_t, in.; cm
ODE	one-dimensional equilibrium
ODF	one-dimensional frozen
ODK	one-dimensional kinetic
O/F	oxidant-fuel ratio, W_{ox}/W_f
P_{C*}	nozzle-throat-inlet total pressure, psia; N/m^2
P_c	chamber pressure measured at injector, psia; kN/m^2
$P_{c,corr}$	total pressure at rocket throat, φP_c, psia; kN/m^2
P_e	static pressure at nozzle exit, psia; kN/m^2
P_0	ambient pressure surrounding engine, psia; kN/m^2
ΔR_{eff}	effective throat radius change, $R_t - R_i$, in.; cm
R_i	initial throat radius measured, in.; cm

R_t throat radius at any time, $\sqrt{\dfrac{W_p \eta_{C^*_K} C^*_{theor}}{\pi g P_{c,corr} C_d}}$, in.; cm

T_c combustion temperature, ^oR

T_w wall temperature, ^oR

V_c chamber volume, in.^{3}; cm^3

V_f fuel injection velocity, $W_f/\rho_f A_f$, ft/sec; m/sec

V_{ox} oxidant injection velocity, $W_{ox}/\rho_{ox} A_{ox}$, ft/sec; m/sec

W_f fuel weight flow, lb/sec; kg/sec

W_{ox} oxidant weight flow, lb/sec; kg/sec

W_p total propellant weight flow, lb/sec; kg/sec

α nozzle divergence angle, deg

ϵ expansion area ratio, A_e/A_t

$\eta_{C_{F,vac}}$ vacuum thrust coefficient efficiency, $\dfrac{\lambda C_m C_{F,d} + \epsilon \dfrac{P}{P_{C^*}}}{C_{F,d} + \epsilon \dfrac{P}{P_{C^*}}}$

$\eta_{C^*_{exp}}$ experimental characteristic exhaust velocity efficiency, $\eta_{I_{sp}} \Big/ \eta_{C_{F,vac}}$

$\eta_{C^*_K}$ constant characteristic exhaust velocity efficiency (determined from calibration firings for each injector)

$\eta_{I_{sp}}$ vacuum specific impulse efficiency, $I_{vac}/I_{vac,theor,equil}$

λ nozzle divergence correction, $(1 + \cos\alpha)/2$

ρ_f fuel density, lb/ft^3; kg/m^3

ρ_{ox} oxidant density, lb/ft^3; kg/m^3

φ momentum pressure loss correction, 0.98

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